NASA-CR-135414

(NASA-CR-135414) SEP ENCKE-87 AND HALLEY RENDEZVOUS STUDIES AND IMPROVED S/C MODEL IMPLEMENTATION IN HILTOP Final Report (Business and Technological Systems, Inc.) 29 p HC A03/MF A01 CSCL 22A G3/12

N78-25105

Unclas 21600

BUSINESS AND TECHNOLOGICAL SYSTEMS, INC.

BTS

BTS-TR-78-55

NASA CR-135414

SEP ENCKE-87 & HALLEY RENDEZVOUS STUDIES
AND
IMPROVED S/C MODEL IMPLEMENTATION IN HILTOP

J. L. Horsewood
F. I. Mann

FINAL REPORT
Part 1 of 2
Contract NAS 3-20950

February 1978

prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center
Cleveland, Ohio 44135

by

BUSINESS AND TECHNOLOGICAL SYSTEMS, INC.
10210 Greenbelt Road, Suite 440
Seabrook, Maryland 20801

SUMMARY

Two studies were conducted to determine the performance requirements for projected state-of-the-art SEP spacecrafts boosted by the Shuttle/IUS to perform a rendezvous with the comet Halley and a rendezvous with the comet Encke during its 1987 apparition. The spacecraft model of the standard HILTOP computer program was assumed. Numerical and graphical results summarizing the studies are presented.

A new, more realistic propulsion system model has been implemented in the HILTOP computer program, in which various thrust subsystem efficiencies and specific impulse are modeled as variable functions of power available to the propulsion system. The number of operating thrusters are staged, and the beam voltage is selected from a set of five (or less) constant voltages, based upon the application of variational calculus. The constant beam voltages may be optimized individually or collectively. A companion document contains the new analysis describing these features, a complete description of program input quantities, and sample cases of computer output illustrating the new program capabilities.

TABLE OF CONTENTS

	Page
SUMMARY	*iii*
I. INTRODUCTION	1
II. SOLAR ELECTRIC PROPULSION COMET RENDEZVOUS STUDIES	2
A. COMET HALLEY	2
B. COMET ENCKE (1987 APPARITION)	12
III. IMPROVED SPACECRAFT MODEL IMPLEMENTATION IN HILTOP	23
IV. REFERENCES	25

I. INTRODUCTION

This document is the final report for Contract NAS 3-20950 and describes work performed for the NASA Lewis Research Center in the field of solar electric propulsion mission analysis.

The report is segmented into two self-contained parts:

(1) Numerical and graphical results summarizing two comet rendezvous studies made with the standard HILTOP computer program [1] and assuming solar electric propulsion combined with the Shuttle/IUS launch vehicle.

(2) An overview of the improvements made to the HILTOP computer program.

II. SOLAR ELECTRIC PROPULSION COMET RENDEZVOUS STUDIES

This section consists of two self-contained parts, each describing the basic assumptions and results of a task to generate performance data for a comet rendezvous mission using solar electric propulsion.

A. Comet Halley. This subsection documents the results of a task to generate performance data for the Halley Comet rendezvous mission giving tradeoffs in delivered mass as a function of variations in launch date, power input to the power processing units and total propulsion system efficiency.

The guidelines of the performance study were as follows:

(1) Neglect throttling effects on thruster performance;
(2) Assume a constant specific impulse of 4770 seconds;
(3) Assume a rendezvous date of December 21, 1985 (50 days before perihelion);
(4) Consider flight times of 1270, 1300, and 1330 days (corresponding to launch dates of June 30, May 31, and May 1, 1982, respectively);
(5) Consider total propulsion system efficiencies of 0.68, 0.70, and 0.72;
(6) Consider thruster subsystems comprised of 6, 7, and 8 operating thrusters of 6.5 kilowatts maximum power each;
(7) Assume a launch vehicle performance corresponding to the Shuttle/IUS; and
(8) Assume a 70 kilowatt array with solar collectors yielding a 3:1 power ratio.

The performance of the Shuttle/IUS is expressed in the form of seven tabular values of launch vehicle payload as a function of hyperbolic excess speed over the range of 3 to 7 kilometers per second. The values of payload provided represented the launch vehicle payload after

subtracting out the IUS adaptor and the SEPS adaptor and therefore may be considered to be the initial spacecraft mass. The SEPS adaptor weight subtracted is variable as it is dependent upon the mass of the spacecraft that it supports. The tabular values were input to a computer program which computes three coefficients - b_1 , b_2 and b_3 - used in the equation for initial spacecraft mass

$$m_\circ = b_1 e^{-v_c/b_2} - b_3$$

such that the sum-square deviations in computed initial mass from the tabular values is minimized. The characteristic speed v_c is a function of the hyperbolic excess speed v_∞ ; i.e.,

$$v_c = \sqrt{v_\infty^2 + v_e^2}$$

where v_e is escape speed from a low altitude circular earth orbit. A value of 11021 m /sec was assumed for v_e . The resulting values of the coefficients are as follows:

$$b_1 = 209698.42 \text{ kg}$$

$$b_2 = 3661.63 \text{ m/sec}$$

$$b_3 = 3757.8797 \text{ kg}$$

The following table presents the tabular values and the corresponding computed values using the above coefficients, both as a function of hyperbolic excess speed.

V_∞ (km/sec)	m_o(tabular) (kg)	m_o(computed) (kg)
3.0	5428	5507
3.5	5163	5157
4.0	4832	4773
4.5	4413	4364
5.0	3949	3937
6.0	3001	3054
7.0	2184	2172

The computed initial spacecraft mass is presented graphically as a function of hyperbolic excess speed in Figure 1.

Tabular values of the 3:1 array power function γ , equal to the ratio of power delivered by the array at a distance r to the power delivered at 1 AU from the sun assuming the array is oriented normal to the sun, represent the best performance estimates currently available. The data exhibit a peak in the power profile at a solar distance of about 1.15 AU, apparently due to temperature effects introduced by the concentrators. Overall, the tabular data indicate substantially higher performance than previously expected from analytical predictions. The 14 tabular values provided were processed using a least square curve fit algorithm to obtain the five coefficients in the equation

$$\gamma = \frac{1}{r^2} \sum_{i=1}^{5} a_i \left(\frac{1}{r^2}\right)^{(i-1)/4}$$

The coefficients thus obtained are as follows

$$a_1 = -31.45129$$

$$a_2 = 198.79617$$

$$a_3 = -404.72935$$

$$a_4 = 351.66829$$

$$a_5 = -113.28382$$

A comparison of the tabulated and computed values of γ as a function of solar distance r is represented in the following table.

r (AU)	γ(tabular)	γ(computed)
1	1.0	1.0
1.3	1.0554	1.0644
1.4	1.0083	1.0039
1.5	0.9543	0.9427
1.6	0.8878	0.8847
1.7	0.8261	0.8308
1.8	0.7742	0.7809
1.9	0.7299	0.7346
2.0	0.6953	0.6917
2.5	0.5166	0.5163
3.0	0.3906	0.3895
3.5	0.2992	0.2965
4.0	0.2271	0.2277
4.5	0.1745	0.1762

The computed power function γ is displayed as a function of solar distance in Figure 2. Superimposed on the graph are three dashed lines representing the maximum values of γ to be employed for 6, 7 and 8 operational thrusters. These maximum values of γ result from the assumptions that the maximum power input to each PPU is 6.5 kilowatts and the maximum array output power at 1 AU is 70 kilowatts. That is, the maximum value of γ permitted for n_{th} thrusters is

SHUTTLE/IUS VEHICLE PERFORMANCE CAPABILITY

SPACECRAFT INITIAL MASS VS. LAUNCH EXCESS SPEED

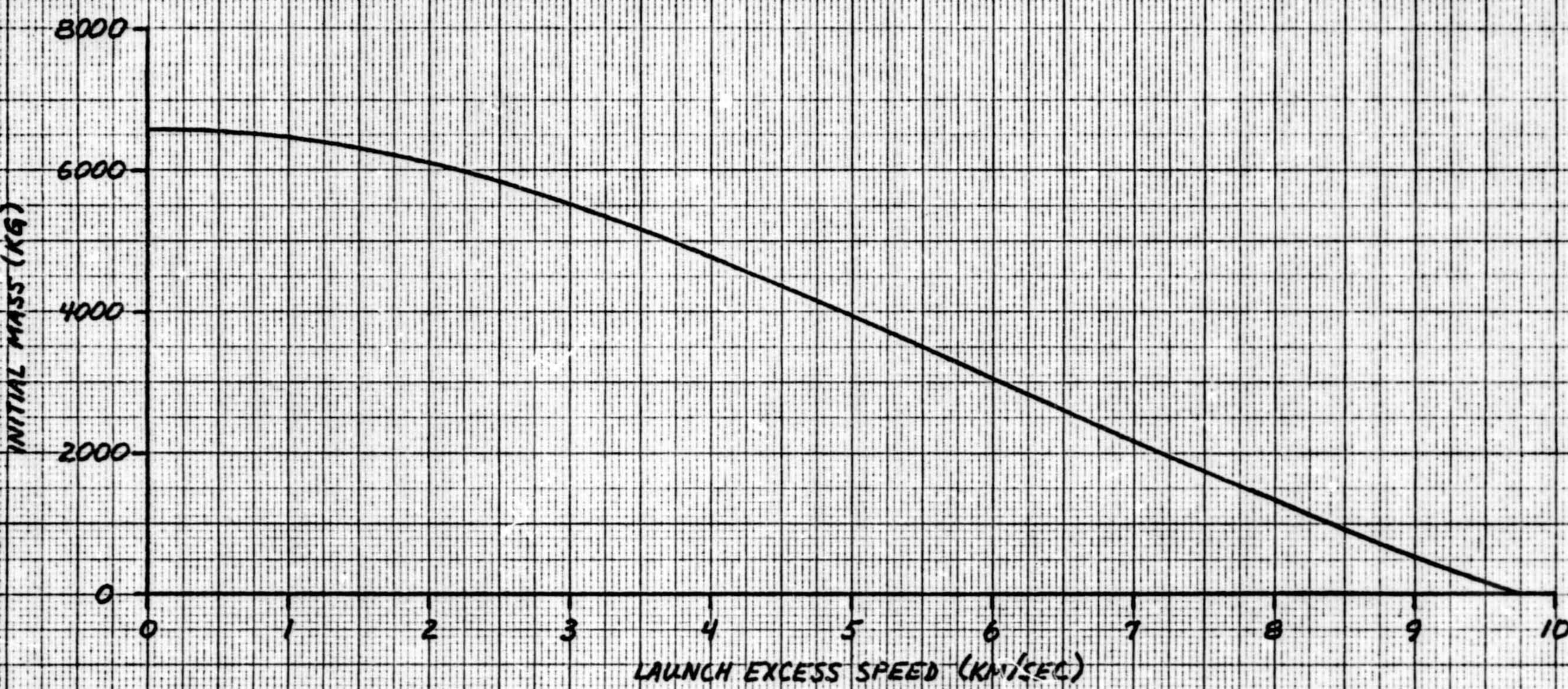


FIGURE 1

SOLAR ARRAY PERFORMANCE CHARACTERISTICS

POWER FUNCTION VS SOLAR DISTANCE

3:1 COLLECTOR

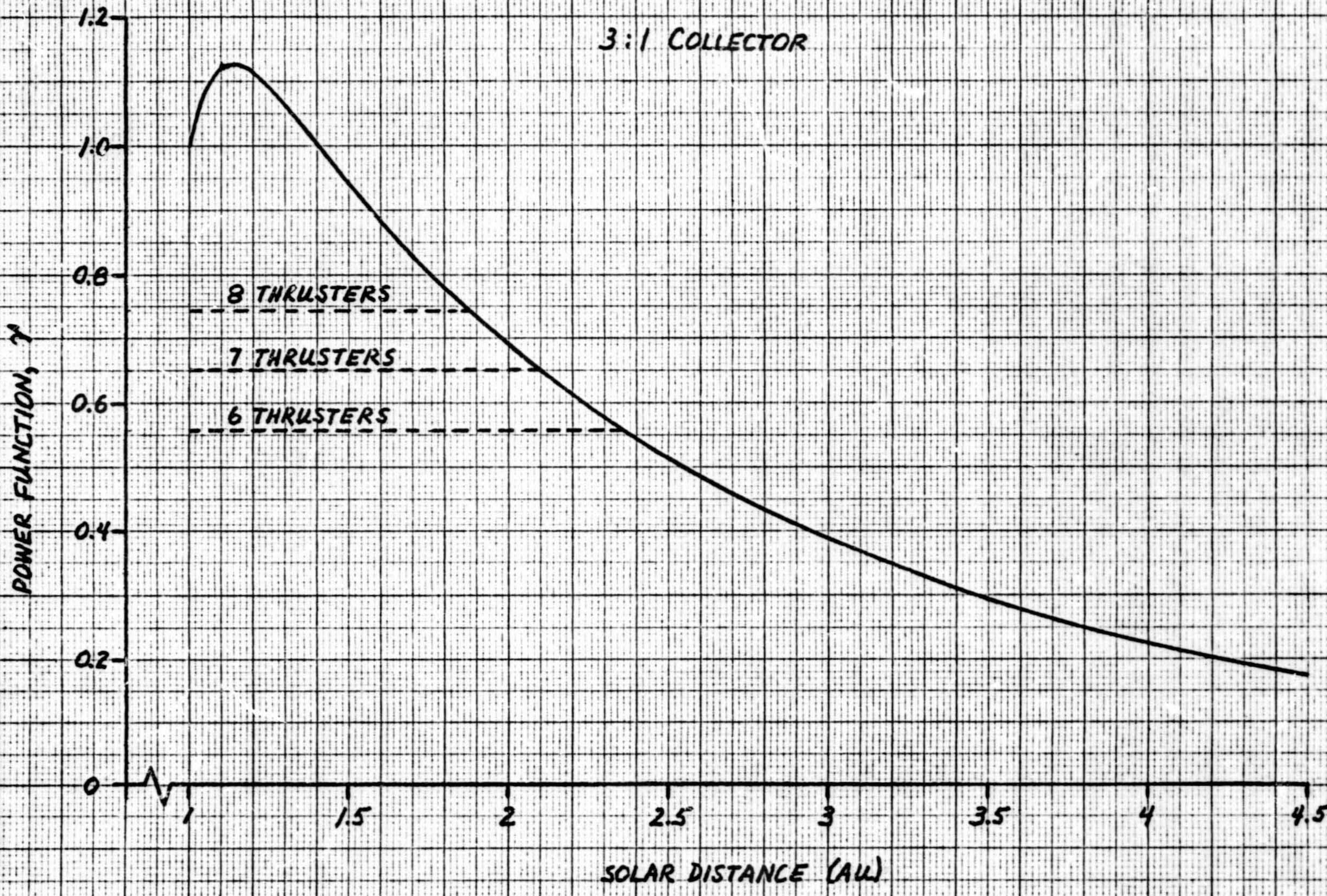


FIGURE 2

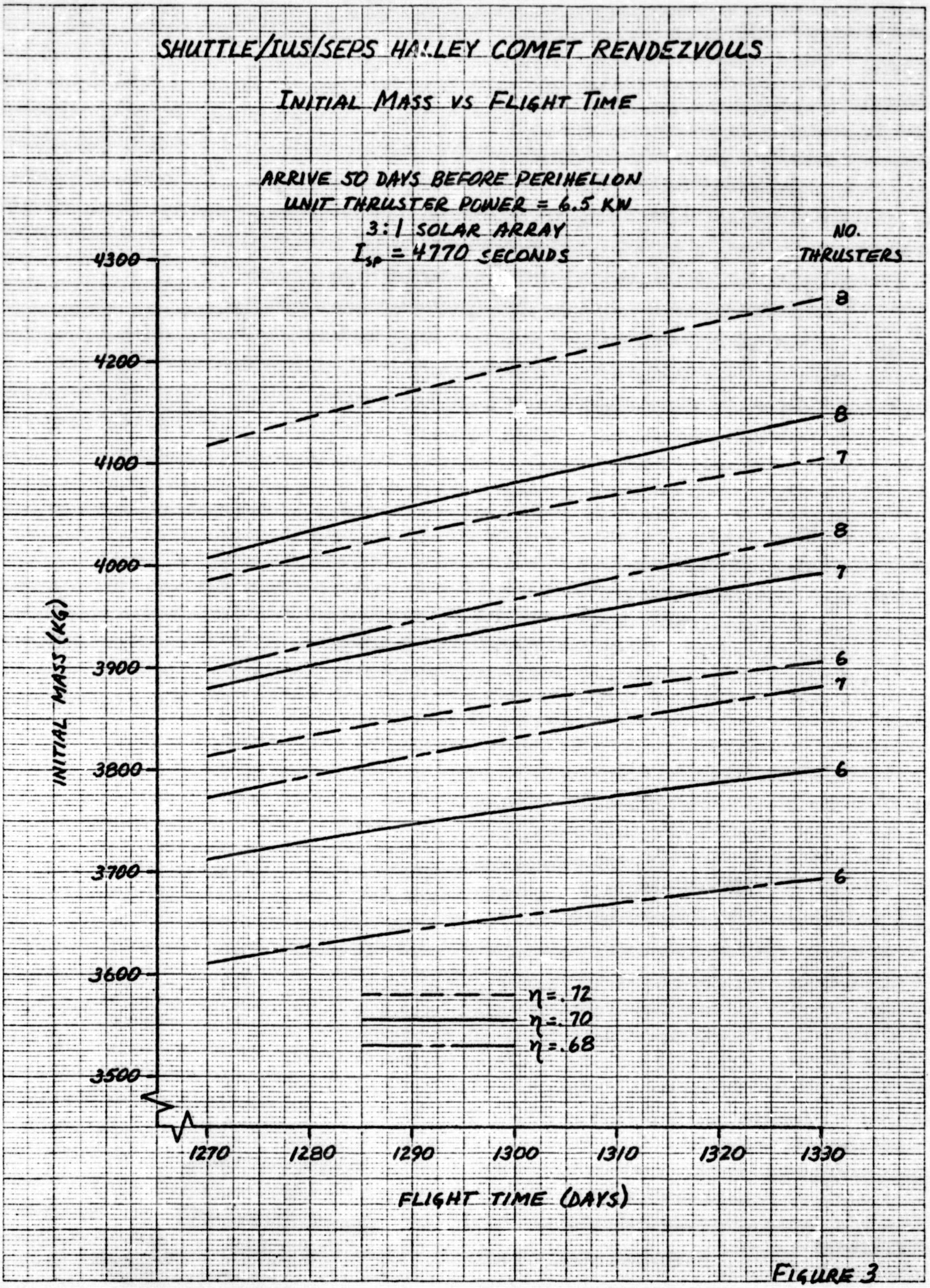

SHUTTLE/IUS/SEPS HALLEY COMET RENDEZVOUS
INITIAL MASS VS FLIGHT TIME
ARRIVE 50 DAYS BEFORE PERIHELION
UNIT THRUSTER POWER = 6.5 KW
3:1 SOLAR ARRAY
I_{sp} = 4770 SECONDS
NO. THRUSTERS
INITIAL MASS (KG)
FLIGHT TIME (DAYS)
η = .72
η = .70
η = .68

FIGURE 3

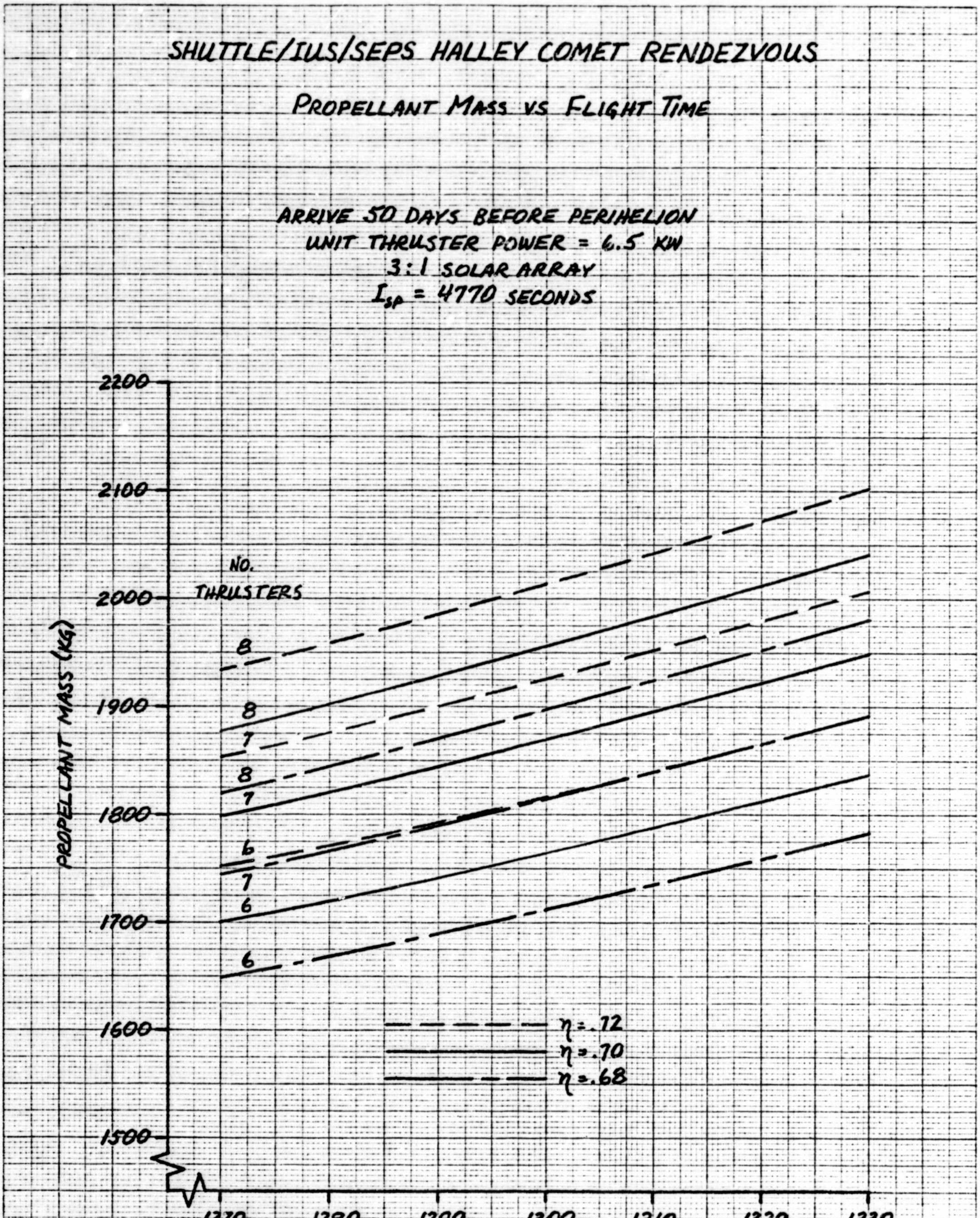

SHUTTLE/IUS/SEPS HALLEY COMET RENDEZVOUS
PROPELLANT MASS VS FLIGHT TIME
ARRIVE 50 DAYS BEFORE PERIHELION
UNIT THRUSTER POWER = 6.5 KW
3:1 SOLAR ARRAY
I_{sp} = 4770 SECONDS
PROPELLANT MASS (KG)
2200
2100
2000
1900
1800
1700
1600
1500
NO.
THRUSTERS
8
8
7
8
7
6
7
6
6
η = .72
η = .70
η = .68
1270
1280
1290
1300
1310
1320
1330
FLIGHT TIME (DAYS)

FIGURE 4

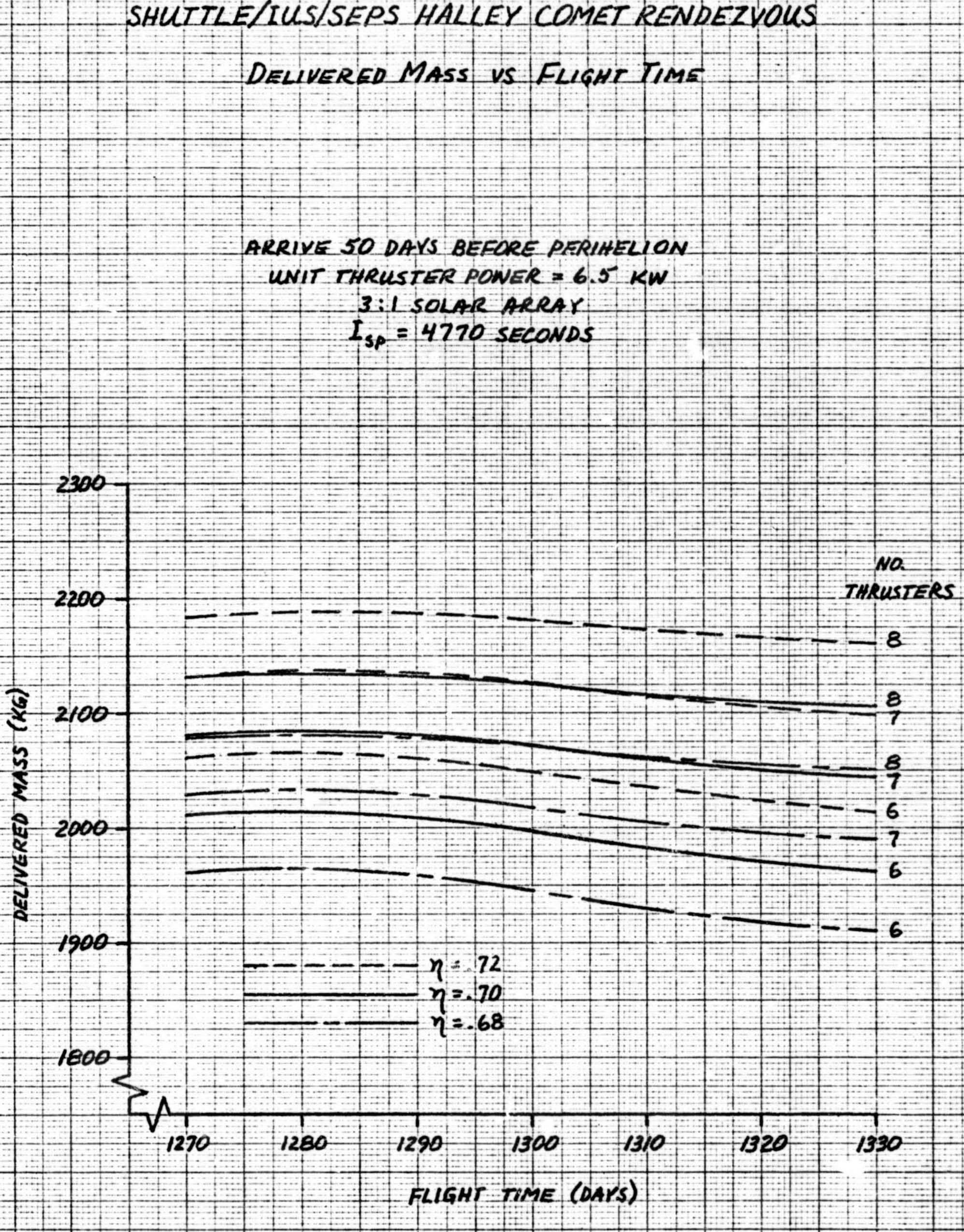

SHUTTLE/IUS/SEPS HALLEY COMET RENDEZVOUS
DELIVERED MASS VS FLIGHT TIME
ARRIVE 50 DAYS BEFORE PERIHELION
UNIT THRUSTER POWER = 6.5 KW
3:1 SOLAR ARRAY
I_{sp} = 4770 SECONDS
DELIVERED MASS (KG)
2300
2200
2100
2000
1900
1800
NO. THRUSTERS
8
8
7
8
7
6
7
6
6
η = .72
η = .70
η = .68
1270
1280
1290
1300
1310
1320
1330
FLIGHT TIME (DAYS)

FIGURE 5

SHUTTLE/IUS/SEPS HALLEY COMET RENDEZVOUS MISSION

Mass Performance Capabilities

Case #	t_f (days)	n_{th}	η	v_∞ (m/sec)	m_o (kg)	m_p (kg)	m_d (kg)
1	1270	6	.68	5372.6	3610.6	1649.3	1961.3
2			.70	5257.1	3712.3	1700.7	2011.5
3			.72	5141.6	3813.4	1752.8	2060.6
4		7	.68	5187.7	3773.1	1744.5	2028.6
5			.70	5065.9	3879.5	1798.8	2080.7
6			.72	4944.2	3985.2	1853.8	2131.4
7		8	.68	5045.3	3897.5	1820.4	2077.0
8			.70	4918.2	4007.7	1876.9	2130.8
9			.72	4790.9	4117.2	1933.9	2183.3
10	1300	6	.68	5320.0	3656.9	1712.2	1944.7
11			.70	5200.8	3761.6	1764.4	1997.2
12			.72	5081.3	3866.1	1816.9	2049.3
13		7	.68	5121.0	3831.4	1814.6	2016.8
14			.70	4994.9	3941.2	1870.0	2071.3
15			.72	4868.1	4050.9	1925.5	2125.4
16		8	.68	4965.0	3967.2	1897.8	2069.4
17			.70	4833.1	4081.1	1955.8	2125.3
18			.72	4700.2	4194.8	2013.9	2180.9
19	1330	6	.68	5278.1	3693.8	1783.2	1910.6
20			.70	5157.1	3799.9	1837.6	1962.3
21			.72	5035.7	3905.8	1892.1	2013.7
22		7	.68	5063.3	3881.8	1891.5	1990.3
23			.70	4934.7	3993.4	1949.3	2044.1
24			.72	4805.3	4104.9	2007.2	2097.7
25		8	.68	4891.5	4030.8	1980.5	2050.3
26			.70	4756.4	4146.8	2041.1	2105.7
27			.72	4620.3	4262.7	2101.7	2161.0
28		6	.70	5243.3	3724.3	1710.5	2013.9
29		7	.70	5045.4	3897.4	1813.2	2084.1
30		8	.70	4891.8	4030.5	1896.3	2134.3

$$\gamma_{max} = 6.5\ n_{th}/70$$

The desired performance data were obtained by generating fully optimized trajectories for the 27 cases comprised of the combination of the three specified values of each of the three variables - flight time, number of operating thrusters, and propulsion system efficiency. Each trajectory was optimized with respect to the thrust direction at each point in time and with respect to the magnitude and direction of the hyperbolic launch excess speed. The optimization objective was to maximize total delivered mass. All cases resulted in continuous burn solutions and with hyperbolic asymptote declinations less than 15 degrees. A somewhat surprising result achieved was that the delivered mass increased with decreasing flight time for a given number of thrusters and propulsion system efficiency, although the peaks appeared to occur in the vicinity of the shortest flight time considered. To more precisely identify these peaks, three additional cases were run for 6, 7 and 8 operating thrusters, all at a propulsion system efficiency of 0.70, in which the launch date was also optimized. The results for the 30 cases are summarized in tabular form on the preceding page. Included in this table are the resultant values of the launch hyperbolic excess speed, v_∞ , the initial mass m_o , the propellant mass, m_p , and the delivered mass, m_d , as a function of the flight time, t_f , the number of operating thrusters, n_{th} , and the total propulsion system efficiency, η . The optimal launch date cases are presented as Case Numbers 28-30. The range of delivered masses obtained over the 30 cases is about 1900-2200 kilograms. The three mass parameters are also presented graphically in Figures 3-5 as a function of flight time for constant values of efficiency and number of operating thrusters.

B. Comet Encke (1987 Apparition). This subsection describes a study conducted to determine the performance requirements for a projected state-of-the-art solar electric propulsion spacecraft boosted by the Shuttle/IUS to perform a rendezvous with the comet Encke during its 1987 apparition. The spacecraft model of the standard HILTOP computer program was assumed.

A total of seventeen optimal trajectory solutions to comet Encke were generated, having the following basic assumptions:

- Mission class - Direct (travel angle <360°),
- Launch vehicle - Shuttle/IUS,
- SEP specific impulse - 2800 seconds, a constant,
- Performance index - maximum delivered mass; initial spacecraft mass equal to sum of net mass and propellant mass,
- Solar power law corresponds to default power curve in HILTOP (MODE=5, GAMMAX=1.DO),
- Reference power - a multiple of 3 kw,
- Efficiency η - constant with time,
- Maximum parking orbit inclination - 32.5 degrees.

The nominal trajectory solution assumed a reference power of 24 kw, an efficiency of η = .58, a flight time of 860 days, and rendezvous at 30 days before perihelion. This flight time and arrival time corresponds to a launch date of February 7, 1985.

Variations from the nominal solution consisted of

- Rendezvousing at 50 days before perihelion with the same February 7, 1985 launch date (i.e., a flight time of 840 days),
- Efficiency η = .60,
- Reference powers of 21, 18, 15, and 12 kw.

Parameters defining the nominal trajectory solution are listed (in addition to the basic assumptions given above):

- Net mass - 1916.8 kg
- Initial mass - 2946.4 kg.
- Propellant mass - 1029.6 kg.
- Departure v_∞ - 6066 m/s(C_3 = 36.8 km^2/s^2)
- Departure asymptote declination - -33.8 degrees
- Parking orbit inclination - 32.5 degrees (at limit)
- Maximum thrust - 1.0139 newtons

- Travel angle (ecliptic) - 276.9 degrees
- Maximum solar distance - 2.988 AU
- Minimum solar distance - 0.952 AU
- Communication distance (arrival) - 1.73 AU
- Communication angle (arrival) 14.0 degrees (Sun-Earth-Encke)
- Solar array degradation time - 264.8 days

A common characteristic of all seventeen optimal trajectory solutions generated is that there are no coast phases; all trajectories assume thrusting operation throughout.

Another common characteristic of all solutions is a launch parking orbit inclination of 32.5 degrees (the maximum assumed allowable). A variation of the nominal solution was generated in which the maximum allowable parking orbit inclination was 48.5 degrees; the resulting optimal solution has a parking orbit inclination of 39.3 degrees, a departure asymptote declination of -39.8 degrees, a departure v_∞ of 6006 m/s, and a net mass of 1921.4 kg, which implies that the net mass penalty associated with the 32.5 degree parking orbit inclination constraint is about one-fourth of one percent, a value which is not significant and which contributes slightly to the conservativism of the study results.

The study results are summarized by Table I, in which three masses are displayed (in kilograms) for each of the seventeen generated solutions; the three masses listed for each solution are, respectively, net spacecraft mass, initial spacecraft mass, and propellant mass.

The trajectory profile for the nominal mission is shown in Figure 6. The trajectory profiles for all seventeen cases generated in this study are very similar to the one displayed in this figure. Tic marks (Fig. 6) denote the direction of the thrust vector. The maximum solar distance for the nominal case is 2.99 AU, and the largest maximum solar distance of all cases in the study is 3.06 AU. In the rendezvous sequence, the spacecraft

approaches the comet nearly head-on from the sunlit side; the comet rushes to attempt to overtake the spacecraft, but the spacecraft thrusts in the direction of the comet's motion to effect rendezvous.

Figure 7 shows the communication angle history and communication distance history for the nominal case; all other cases are very similar in communication history to the nominal case, and, of course, identical in the post-rendezvous phase. The communication angle is defined as the angle subtended at the Earth by the line segment between the sun and the spacecraft.

Figure 8 displays the power profiles for the nominal case (24 kw) and for the variation cases of 18 kw and 12 kw reference power. Due to the high similarity of the trajectory profiles, these power curves are essentially scaled versions of each other. The power curves are used to determine thruster staging times (e.g., assuming each thruster operates at a maximum of three kilowatts), the total number of "thruster-hours", and finally the number of operational hours per thruster assuming all available thrusters (excluding spares) operate for the same amount of time. The power curve analysis is summarized by the following table:

Reference Power (kw)	Total Thruster Hours	Operational Hours per Thruster
12	38,232	9,558
18	59,304	9,884
24	71,472	8,934

Each trajectory in the study moves closer to the sun during the first month after launch; the minimum solar distances are displayed in Figure 9. The missions having shorter flight times (toward the "end" of the launch window) do not dip toward the sun just after launch as much as those having longer flight times; this alleviates the initial thermal load, although of course the spacecraft may be thermally designed to withstand temperatures at Encke's perihelion (.337 AU).

TABLE I

ENCKE 1987 RENDEZVOUS

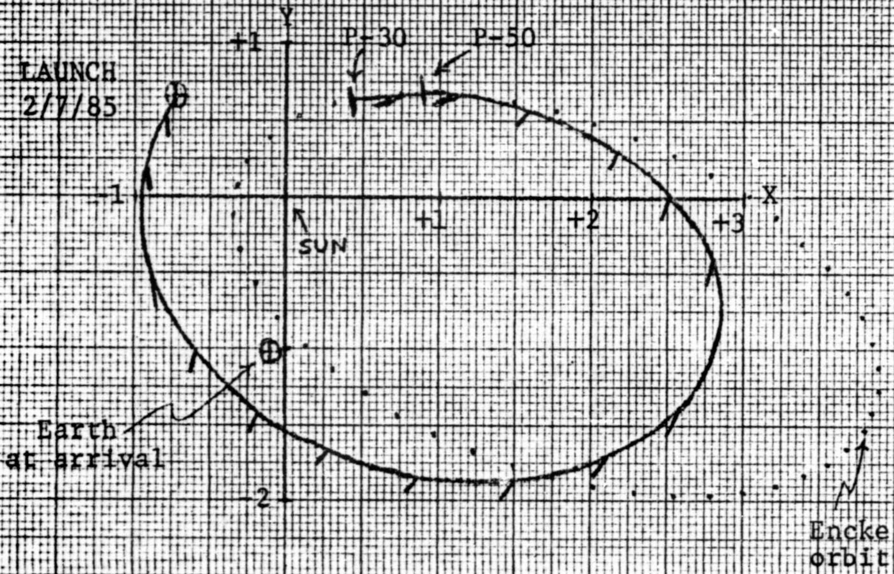
Shuttle/IUS/SEP
Arrive 30 Days Before Perihelion
Efficiency = .58

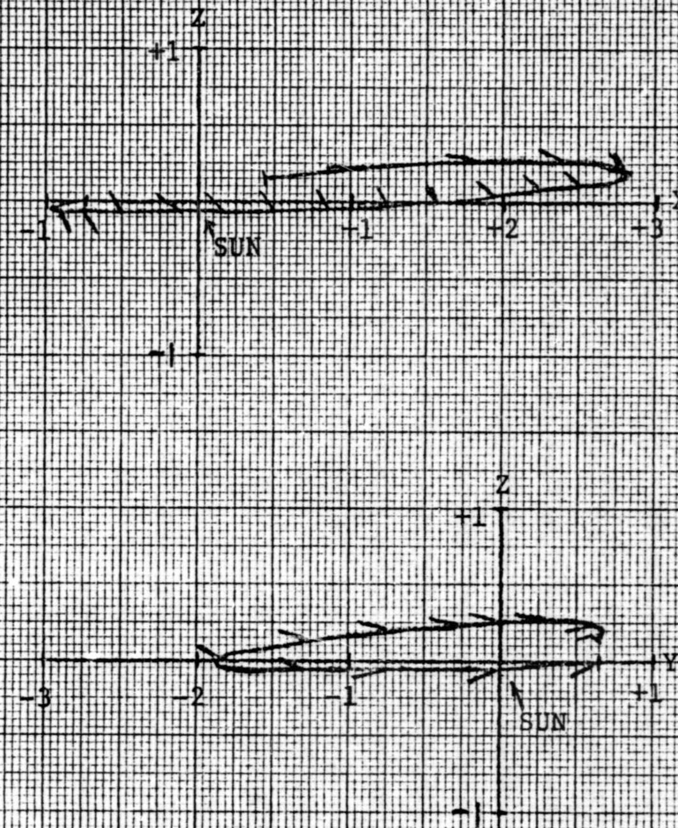
FLIGHT TIME (DAYS)	REFERENCE POWER (KW) 12	15	18	21	24
830	net mass initial mass propellant mass	KEY (masses in kg)	1448.2 2156.8 708.5	1674.0 2501.3 827.3	1893.9 2840.4 946.5
860	1021.6 1532.9 511.3	1256.8 1897.1 640.3	1484.3 2253.9 769.7	1704.3 2603.8 899.5	*1916.8 2946.4 1029.6
890			1466.1 2293.1 827.0	1678.2 2645.7 967.4	1882.6 2991.1 1108.5
860	Variation: Efficiency = .60		1530.4 2326.9 796.5	1756.3 2687.1 930.9	1974.0 3039.6 1065.6
840	Variation: Arrive 50 days before perihelion		1267.3 1987.5 720.2	1460.3 2302.1 841.8	1648.2 2612.1 963.9

* Nominal solution

Figure 6 1978 ENCKE RENDEZVOUS TRAJECTORY PROFILE

Arrival at 30 days before perihelion
Flight time 860 days
Launched by Shuttle/IUS
Specific impulse 2800 sec.
Efficiency .58
Reference Power 24 kw





Note: Distances in AU
Tic marks show thrust direction

Figure 7 1987 ENCKE RENDEZVOUS COMMUNICATION HISTORY

NOMINAL CASE

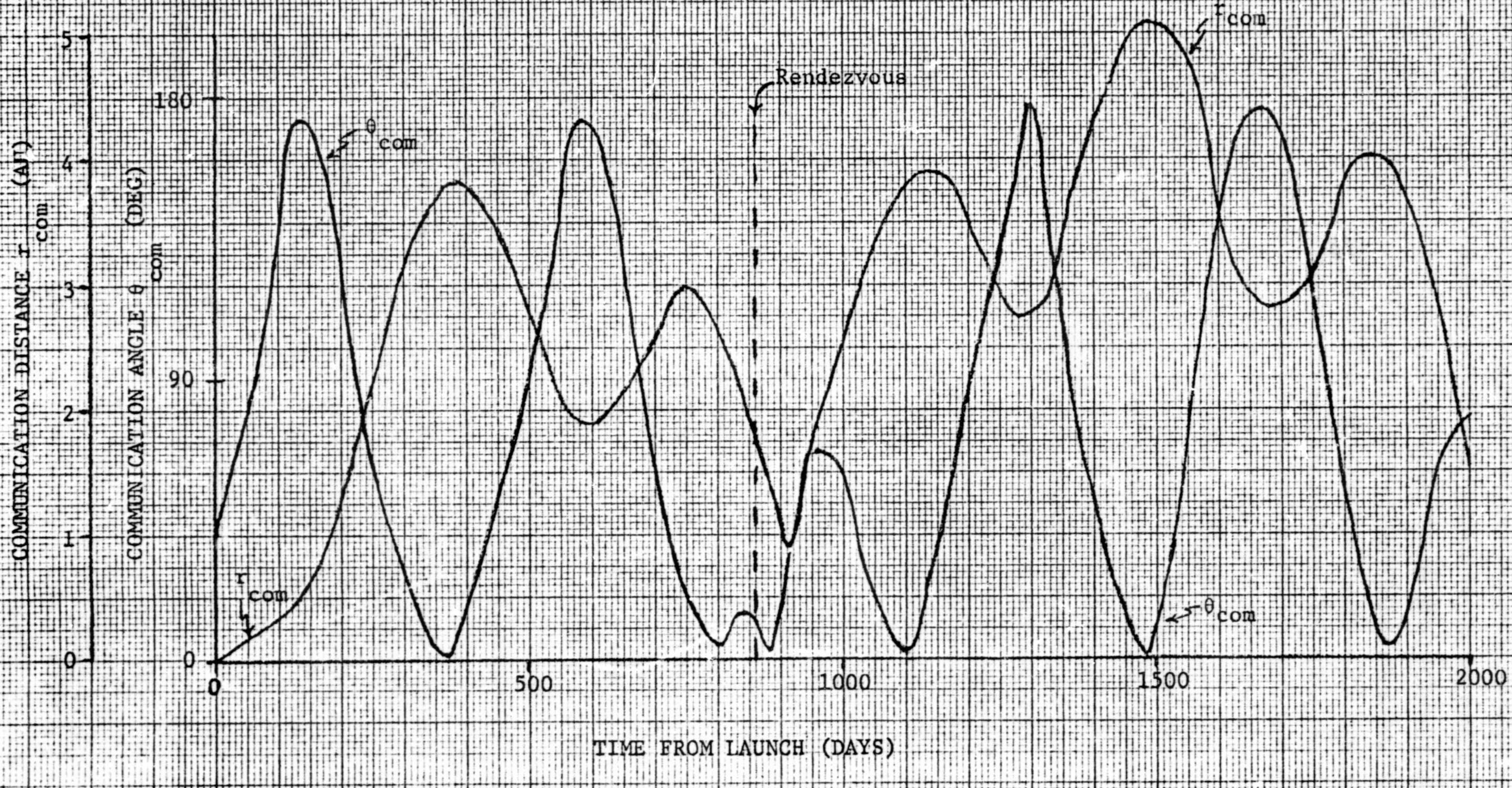
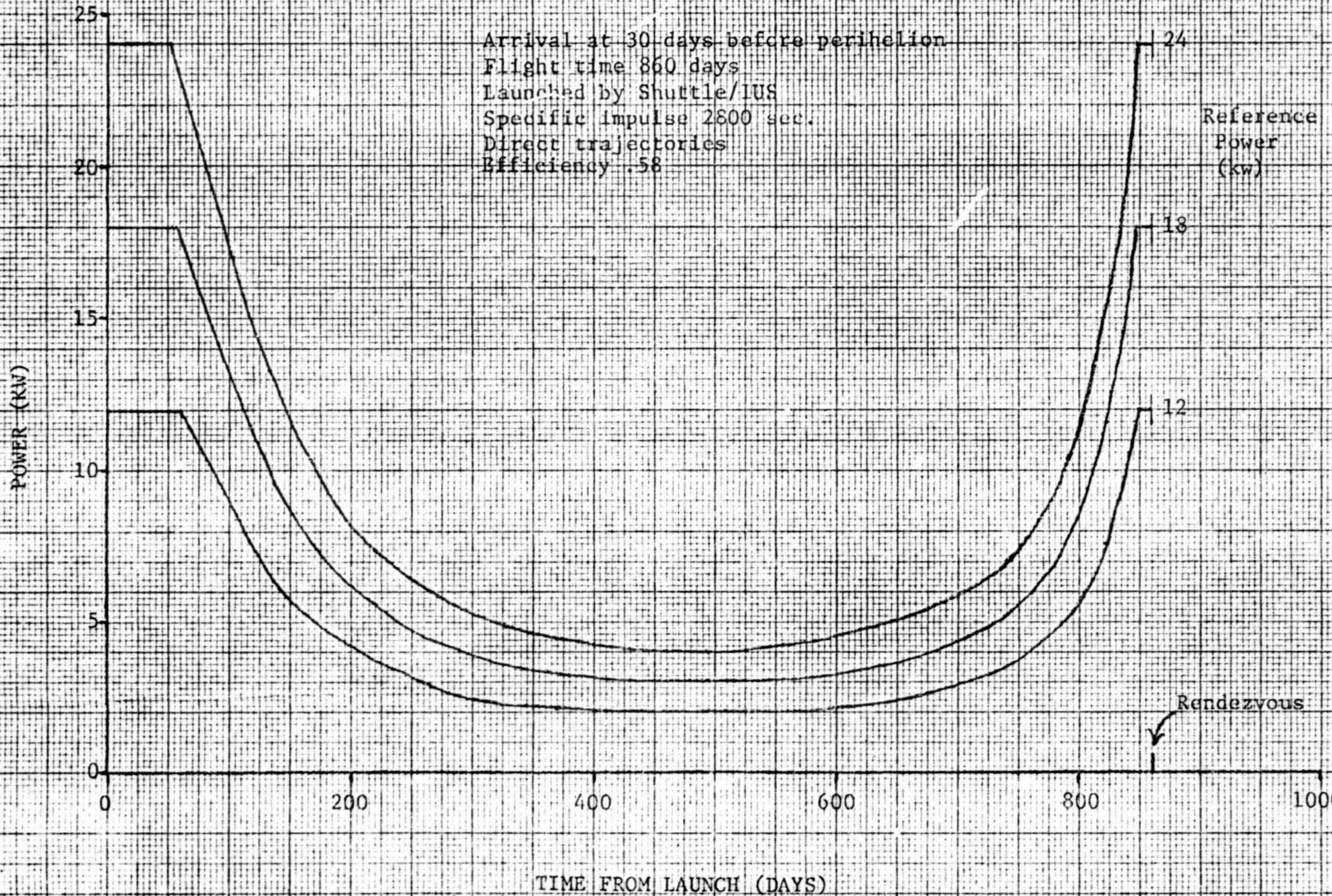
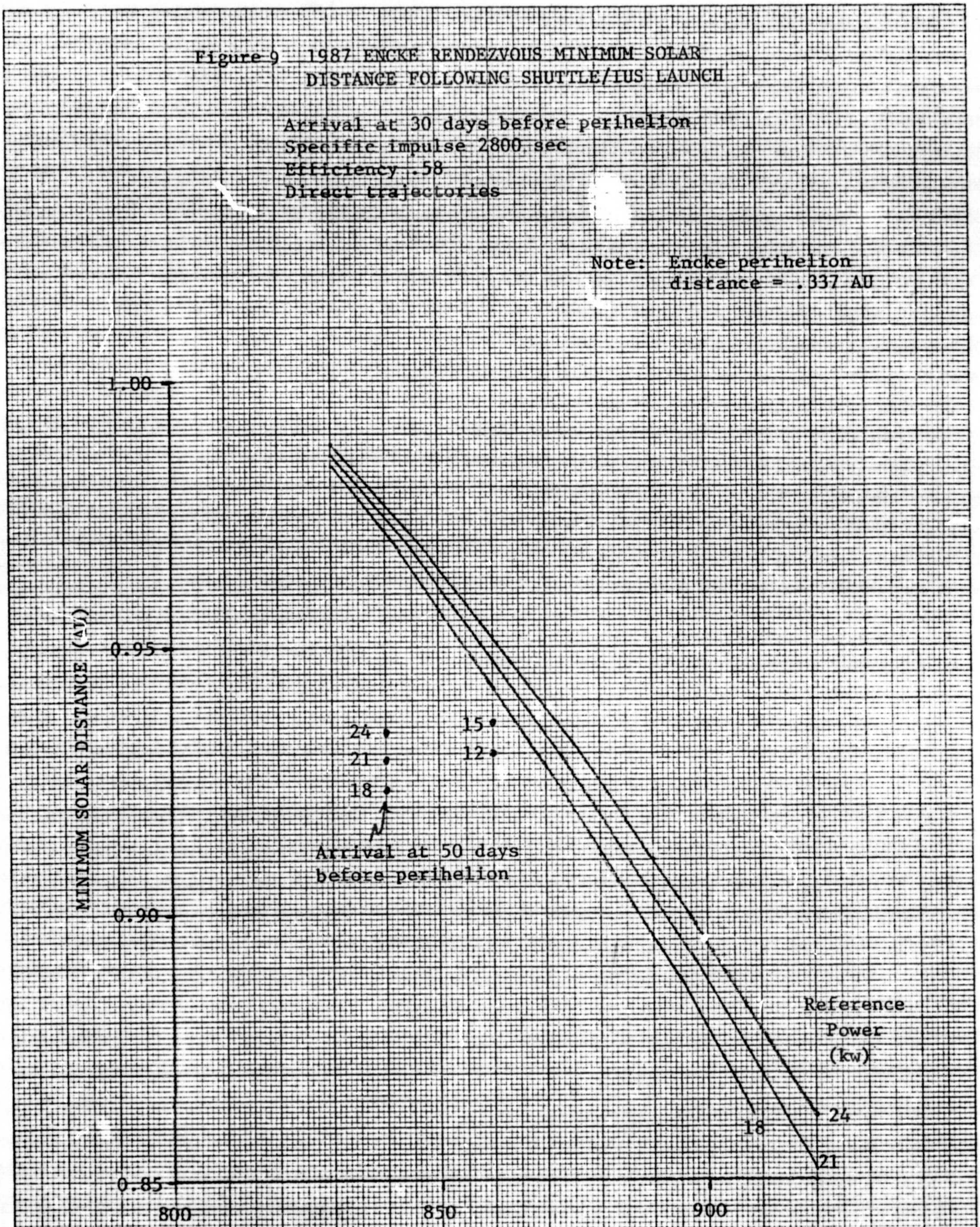


Figure 8 1987 ENCKE RENDEZVOUS POWER PROFILES





ORIGINAL PAGE IS
OF POOR QUALITY

Figure 10 1987 ENCKE RENDEZVOUS LAUNCH-VEHICLE-INDEPENDENT INITIAL-MASS CURVES

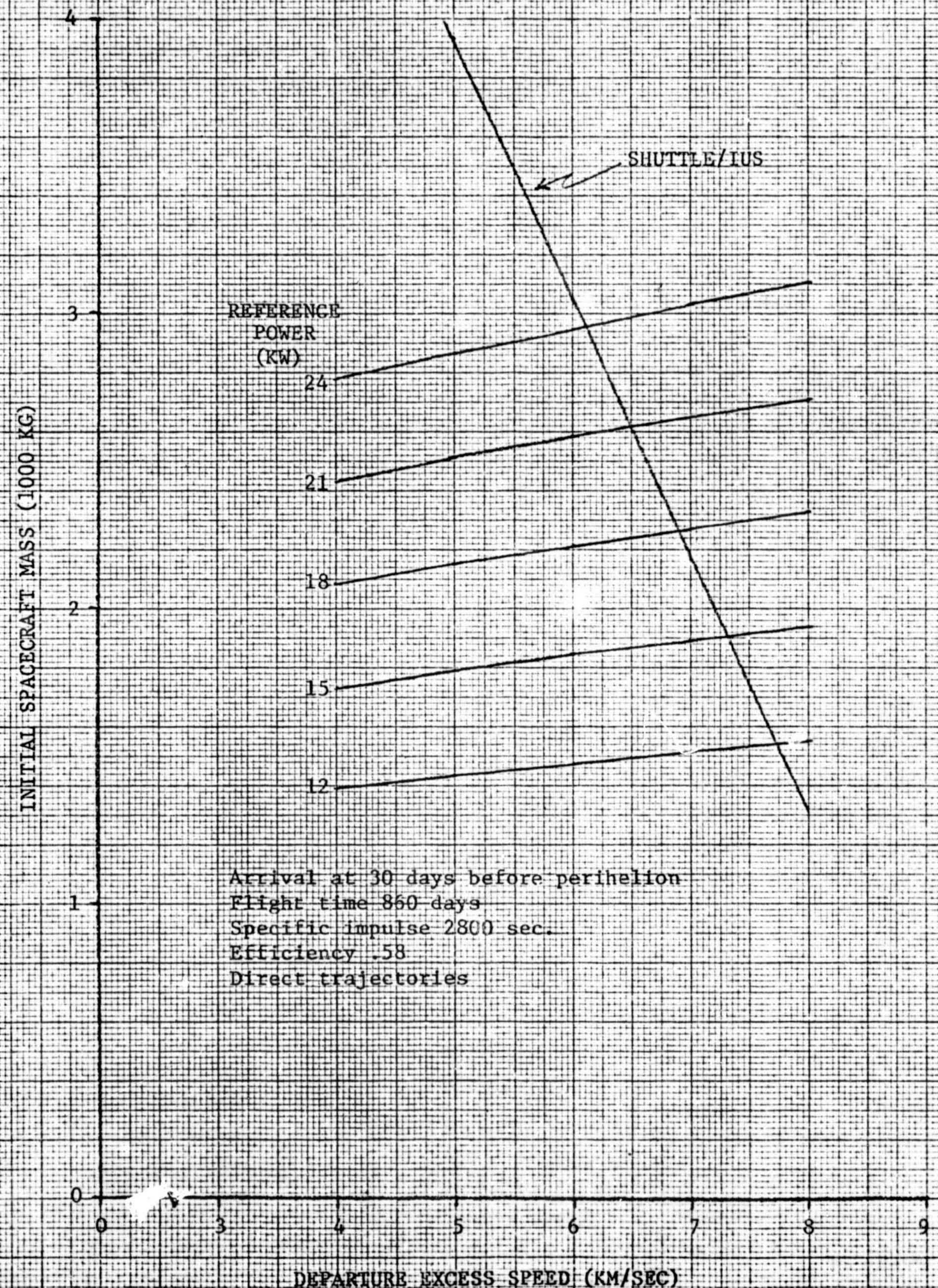


Figure 10 shows initial mass capabilities for the 1987 Encke rendezvous mission generated by HILTOP using the Launch Vehicle Independent mode of operation. The performance curve for the Shuttle/IUS is superimposed. The intersection of a launch vehicle performance curve with a curve of constant reference power determines the initial spacecraft mass and departure hyperbolic excess speed associated with that launch vehicle/reference power combination for the mission.

III. IMPROVED SPACECRAFT MODEL IMPLEMENTATION IN HILTOP

A companion document [2] is the first supplement to the currently existing primary HILTOP program document (published in December 1974; see reference [1]) and describes the modifications and improvements made to the HILTOP electric propulsion trajectory optimization computer program up through February 1978.

A new, more realistic propulsion system model involving the actual ion beam current and voltage has been implemented in the program. The power processor efficiency, ion thruster efficiency, and thruster specific impulse are modeled as variable functions of the (solar array, nuclear, or other) power available to the propulsion system. The number of operating thrusters are staged, and the beam voltage is selected from a set of five (or less) constant voltages, based upon the application of variational calculus. The minimum and maximum number of operating thrusters, the minimum throttling ratio, and the maximum input power to an individual thruster are specified as input data. The constant beam voltages may be optimized individually or collectively.

The new propulsion system logic is activated by a single program input key; program modifications have been designed to retain the "old" HILTOP program within the framework of the new logic, so that old input data files (with no modifications required) will run the new program version and produce identical results as before.

The capability of simulating solar array degradation with the new spacecraft model is not included in this program version; also not included is the capability of simulating the new spacecraft model under the Launch Vehicle Independent (LVI) mode. The simulation of array degradation and the LVI mode remain available with the old spacecraft model.

The execution step requirements of the new program version are a little less than 390K. This compares to 350K for the old version.

The companion report [2] contains the new analysis describing these features, a complete description of program input quantities, and sample cases of computer output illustrating the new program capabilities. A more detailed understanding of optimal electric propulsion engine-control switching time-histories will become available as the new program capabilities are exercised in future studies.

IV. REFERENCES

[1] F. I. Mann and J. L. Horsewood, "Program Manual for HILTOP, A Heliocentric Interplanetary Low Thrust Trajectory Optimization Program," Analytical Mechanics Associates, Inc., Report No. 74-34, December 1974.

Part I - User's Guide (287 pages)

Part II - Subroutine Descriptions (854 pages)

[2] F. I. Mann and J. L. Horsewood, "HILTOP Supplement - Heliocentric Interplanetary Low Thrust Trajectory Optimization Program - Supplement #1," Business and Technological Systems, Inc., Report No. BTS-TR-78-54, February 1978.